FLIGHT DYNAMICS ASPECTS OF THE GRACE FORMATION FLYING

Michael KIRSCHNER
Oliver MONTENBRUCK
Srinivas BETTADPUR

Deutsches Zentrum für Luft- und Raumfahrt
German Space Operations Center (GSOC)
D-82234 Wessling
michael.kirschner@dlr.de

ABSTRACT *– The GRACE mission, which is scheduled for launch at the end of 2001, will be among the first missions ever to exercise close formation flying for improved science data collection. The joint US-German project aims at the exploration of temporal variations of the Earth's gravity field and the investigation of the Earth's atmosphere. Key payload elements comprise a GPS receiver, a high-precision accelerometer and a star sensor onboard each spacecraft as well as a K-band inter-satellite ranging system. Both spacecraft will fly at a 300-500 altitude with a relative along-track separation of about 220±50 km (or 30±6 secs) to build up a gradiometer that is particularly sensitive to high-order harmonic components in the Earth's gravity field. From an operational point of view, the build-up and control of the formation GRACE poses various new requirements, which have been studied at the mission control center (GSOC): the safe injection of the two spacecraft from the launcher, the control of the semi-major axes of both spacecrafts to maintain the inter-satellite separation within tolerable margins and the accurate maintenance of the line-of-sight pointing of the K-band radar.*

KEYWORDS: GRACE, formation, SGP4, upper stage separation.

INTRODUCTION

In 1997, the Gravity Recovery and Climate Experiment (GRACE) mission was selected by NASA for development under a new Office of Mission to Planet Earth program called Earth System Science Pathfinder (ESSP). As an innovation, the principal investigator and the mission team are ultimately responsible for developing the flight mission hardware from selection to a launch-ready condition, with minimal direct NASA oversight, for accomplishing the scientific objectives and delivering the proposed measurements to the broader Earth science community and general public as expediently as possible [1].

The gravity field of the Earth is variable in both space and time. The primary objective of the GRACE mission is to obtain accurate global models for the mean and time variable components of the Earth's gravity field [2]. The primary product of the GRACE mission is a new model of the Earth's gravity field every 15 to 30 days for a period of five years. This will be achieved by making accurate measurements of the inter-satellite range change between two co-planar, low altitude, near-polar orbiting satellites, using a K-Band microwave tracking system. In addition, each satellite will carry a geodetic quality Global Positioning System (GPS) receiver and a high accuracy accelerometer to enable accurate orbit determination, spatial registration of gravity data, and the estimation of gravity field models. The Earth gravity field estimates obtained from data gathered by the GRACE mission will provide, with unprecedented accuracy, integral constraints on the global mass distribution and its temporal variations within the Earth system.

The GRACE satellites will be launched on-board a Eurockot launch vehicle from Plesetsk, Russia, between February 27 and end of March, 2002. Both satellites will be placed in the same nominal circular orbit of 500 km at an inclination of 89 deg. Following Launch and Early Orbit Phase (LEOP) operations, the orbits of the two satellites will evolve naturally for the remainder of the mission. During the science data collection, the two GRACE satellites will point their K-Band feed horns towards each other to a high precision. Over the mission lifetime the two satellites will remain in coplanar orbits. Due to differential drag force, the along-track separation will vary, and station-keeping maneuvers will be required to keep the two satellites within 170 to 270 km of each other. To minimize the interruption of science data collection, the time between two orbit maintenance maneuvers should be at least 30 days.

S/C SEPARATION FROM UPPER STAGE AND ORBIT ACQUISITION

While common telecom constellations require an in-plane separation of the associated spacecraft that makes up a substantial fraction of an orbit, the two GRACE satellites need only be separated by about 200-250 km to achieve their target formation. It has therefore been decided to eject both satellites simultaneously and to perform no intermediate re-ignition of the upper stage. This concept is further assisted by the three-axis attitude stabilization of the Breeze upper stage, which allows a controlled ejection of the two GRACE satellites in any desirable direction.

The ejection mechanism separates both satellites using a velocity increment of $\Delta v=\pm 0.25 \div 0.30$ m/s perpendicular to the symmetry axis of the upper stage. When applied in the along-track direction, the semi-major axes of GRACE 1 and GRACE 2 differs by roughly 1 km, which results in a continuous increase of the along track separation of 10 km per orbit or 160 km/d. The nominal separation would thus be obtained in about 1.5 days, which leaves too little time for attitude acquisition, satellite checkout, and preparation of a drift stop maneuver. To cope with this difficulty, a pitch angle of 30° is applied to the upper stage (Fig. 1), which reduces the along-track velocity increment and the resulting drift to one half of the value given above. The minimum available time for reaching the nominal separation of 220 km [2] can thus be extended to three to four days.

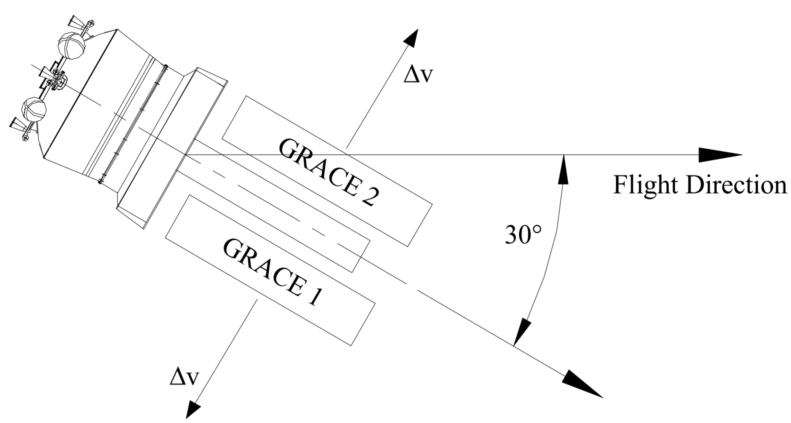

Fig. 1. Orientation of the Breeze upper stage before separation of the two GRACE satellites. To adjust the along-track component of the separation velocity, a pitch angle of 30° is applied to the stage.

The radial component of the separation velocity results in an eccentricity difference and an associated cycloidal relative motion of the two spacecraft, which is depicted in Fig. 2. Since the s/c ejected in the backward and inward direction (“1”) has a smaller semi-major axis, its mean motion is larger than that of the upper stage and the other s/c (“2”). Accordingly, its backward drift is quickly reverted and the s/c overtakes after several minutes. The relative distance increases monotonically for about 80 mins, at which time the spacecraft have reached a separation of approximately 5 km from each other.

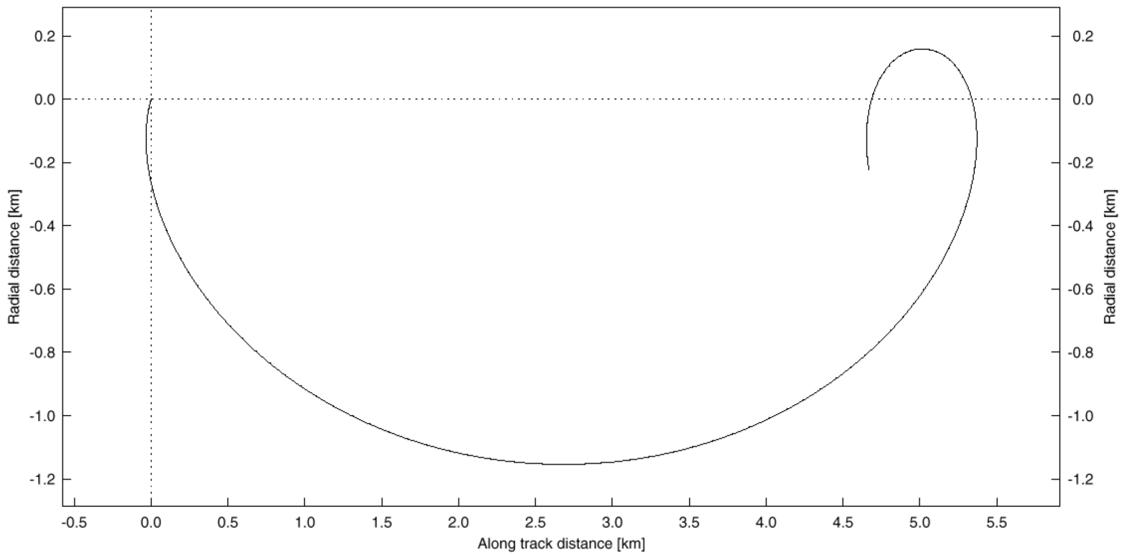

Fig. 2. Relative motion of the GRACE spacecraft after separation

150 seconds after separation, the Breeze upper stage performs a small maneuver to leave the GRACE formation. To minimize the collision risk, a de-boost maneuver about 15 min after separation will follow to generate a final Breeze orbit with a perigee altitude below 150 km and to guarantee a rapid decay of the apogee.

The drift has to be stopped at the latest, when nominal separation is reached. This can either be done by one drift stop maneuver after about three days or by a sequence of drift stop maneuvers, which could extend the complete drift time. In any case, the drift stop maneuvers are executed to remove the relative drift in the along-track direction. In order to reduce workload during LEOP, the stop maneuver will be performed with only one satellite. To this end the along-track velocity has to be increased by $\Delta v \sim 0.3$ m/s if the leading s/c (“1”) performs the maneuver, or the speed is reduced by the same amount if the trailing s/c is used for the maneuver. At a thrust level of 80 mN and a weight of 450 kg the total maneuver takes about 25 mins.

FORMATION KEEPING

Once the initial formation has been achieved, the orbits of both spacecraft evolve naturally under the action of the gravitational and non-gravitational forces. During short time scales (about 1 rev), the relative motion is governed by the differences in eccentricity (which are confined to less than $1.4 \cdot 10^{-4}$ as a consequence of the separation and drift stop strategy) and the 1.9° difference in eccentric anomaly (scaled by an eccentricity of less than $2{,}5 \cdot 10^{-3}$). For longer time scales, the difference Da of the mean semi-major axes results in a change

$$\frac{dL}{dt} = a \cdot \Delta n = -\frac{3}{2} n \cdot \Delta a \qquad (1)$$

of the along-track separation L by roughly $150 \cdot \Delta a$ per day. Accordingly, the semi-major axis difference must be controlled within a boundary of ±25 m, to stay within the assigned distance limits over at least one month.

Under the action of the atmospheric drag the altitude of both spacecraft decreases from an initial value of 500 km to a lower limit of 300 km throughout the mission life-time. In parallel to the secular decrease, a difference in the relative semi-major axes builds up due to the non-equality of the aerodynamic forces acting on the two satellites. Despite the symmetrical design of the satellites, their ballistic coefficients ($B=c_D \cdot A/m$) differ as a consequence of a slightly different attitude (cf. Fig. 3). To ensure a line-of-sight orientation of the K-band link, pitch angles between 0.4° and 2.2° are required depending on the altitude and relative distance (cf. Table 1). Within the nominal operations regime, the associated ballistic coefficient difference ranges from 0.14% to 0.32%, if the mass of both spacecraft is identical. In this case the leading satellite always has a higher drag and, as a consequence, a higher decay rate [3, 4, 5].

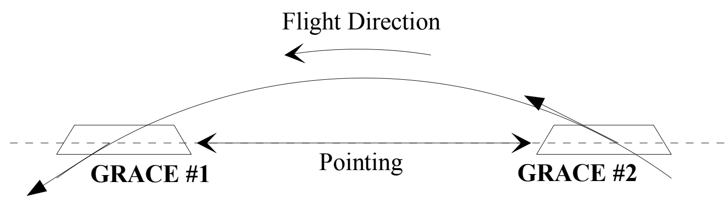

Fig. 3. On-orbit constellation of the GRACE satellites

Table 1. Pitch angle depending on separation and altitude (bold values apply for the lower and upper boundary of nominal separation)

Separation [km]	100	**170**	**270**	500
H = 300 [km]	0.43	**0.73**	**1.16**	2.15
H = 500 [km]	0.42	**0.71**	**1.12**	2.08

The different ballistic coefficients result in a rate of change

$$\frac{d\Delta a}{dt} = -\Delta B \cdot \rho \cdot v^2 \frac{1}{n} = -\Delta B \cdot \rho \cdot a^2 n \tag{2}$$

of the semi-major axis difference (cf. [6]), where v~7.5 km/s is the orbital velocity of the satellites and ρ denotes the atmospheric density. Insertion of eqns. (1) and (2) then yields a differential equation

$$\frac{dL}{d\Delta a} = \frac{3}{2} \frac{1}{\Delta B \rho a^2} \Delta a \tag{3}$$

for the variation of the mutual distance of the two spacecraft as a function of the semi-major axis difference. Under the assumption of a constant atmospheric density its solution is given by the relation

$$L(t) = L_0 + \frac{3}{4} \frac{1}{\Delta B \cdot \rho \cdot a^2} (\Delta a(t))^2 \tag{4}$$

i.e. the separation varies with the square of the semi-major axis difference. In order to maximize the time between subsequent formation keeping maneuvers, the two spacecraft should initially be placed at the maximum desirable separation with a semi-major axis offset

$$\Delta a_{\max} = \sqrt{4/3 \cdot \Delta B \cdot \rho \cdot a^2 \cdot (L_{\max} - L_{\min})} \tag{5}$$

of the leading spacecraft. Here L_{max}-L_{min} is the width of the station keeping box, i.e. the difference of the maximum and minimum separation. During the station keeping cycle, a parabolic profile of relative altitude versus relative separation is obtained under the assumption of constant drag. The minimum distance is achieved, when both semi-major axes are equal. Thereafter, the separation increases again to its maximum value, at which time the semi-major axis of the leading spacecraft is smaller than that of the trailing s/c by Δa_{max}. At the end of the station keeping cycle a small maneuver is performed to raise the semi-major axis of the leading s/c or, alternatively, to lower the semi-major axis of the trailing satellite. For given density and aerodynamic properties, the length of the station keeping cycle, i.e. the time between subsequent correction maneuvers, is given by

$$t_{\mathrm{cyc}} = \sqrt{\frac{16}{3} \frac{(L_{\max} - L_{\min})^2}{\Delta B \cdot \rho \cdot a^2 n^2}} \tag{6}$$

Depending on the actual difference between the ballistic properties of both GRACE satellites, the mission altitude and the solar activity, it is currently foreseen to perform formation keeping maneuvers every 30 to 180 days. To allow for an uninterrupted complete coverage of the surface of the Earth, the separation between consecutive maneuvers should always be a multiple of about 15 days. While the separation is allowed to vary between 170 and 270 km, the mean semi-major axis difference varies by as little as 20 m during a single correction cycle.

For the first cycle after separation from the Breeze upper stage, Table 2 shows a comparison between the analytical prediction based on eqns. (5) and (6) and a numerical simulation (cf. Fig. 4). The maneuver size for this Δa_{max} is 0.86 cm/s, the corresponding fuel is about 6.6 g and the maneuver duration is about 50 sec. Even though the principal characteristics of the relative motion are well explained by the simplistic model derived above, the ideal parabolic motion in phase space is notably perturbed as soon as realistic density variations are taken into account (cf. Fig. 4). In addition, the difference in the ballistic coefficient ΔB will also be influenced by changes in the s/c mass due to different attitude activities. As a consequence, the available station keeping box will not be fully exploited to leave some margin for unpredictable density variations. Throughout the mission, the relative distance and semi-major axis difference will be monitored on a routine basis as well as the evolution of the mass of both s/c and, if necessary, the deadband for the attitude activities (attitude thruster fuel consumption) will be changed for proper ΔB balance. In parallel, predictions of the expected motion in the L-Δa-plane will be carried out to decide on a suitable maneuver time and size. These predictions will make use of forecasted solar flux values provided by the European Space Operations Center (ESOC) on a daily and monthly basis.

Table 2. Comparison of predicted and simulated drift cycle

	ΔB [m^2/kg]	ρ [kg/m^3]	a [km]	n [1/s]	ΔL_{max} [km]	Δa_{max} [m]	t_{cyc} [d]
Prediction	$1.56 \cdot 10^{-5}$	$2.35 \cdot 10^{-12}$	6878	0.0011	100	15.3	185
Simulation	variable	variable	variable	variable	90	15.5	174

For proper maneuver planning, the individual mean semi-major axes must be resolved from the osculating states with an accuracy of better than 1m. While a purely numerical averaging requires lengthy time intervals to give proper results, analytical models do not, in general, provide sufficient accuracy. It has therefore been decided to use a suitable combination of both techniques to derive the mean semi-major axes of the GRACE spacecraft at any time. To this end, the trajectories are numerically integrated over a time interval of six orbits using a rigorous force model and initial conditions from the latest orbit determination. Using a sampling interval of 10 mins, the resulting state vectors y are fit to the SGP4 model by the least square method [7, 8]. The SGP4 orbit model, which forms the basis of the NORAD twoline element sets [9], is based on the analytical theory of Brouwer and accounts for the Earth gravitational field through zonal terms $J2$, $J3$, $J4$, and the atmospheric drag through a power density function assuming a non-rotating spherical atmosphere. Using this technique, smooth Da values with an uncertainty of less than 1m are obtained (cf. Fig. 4).

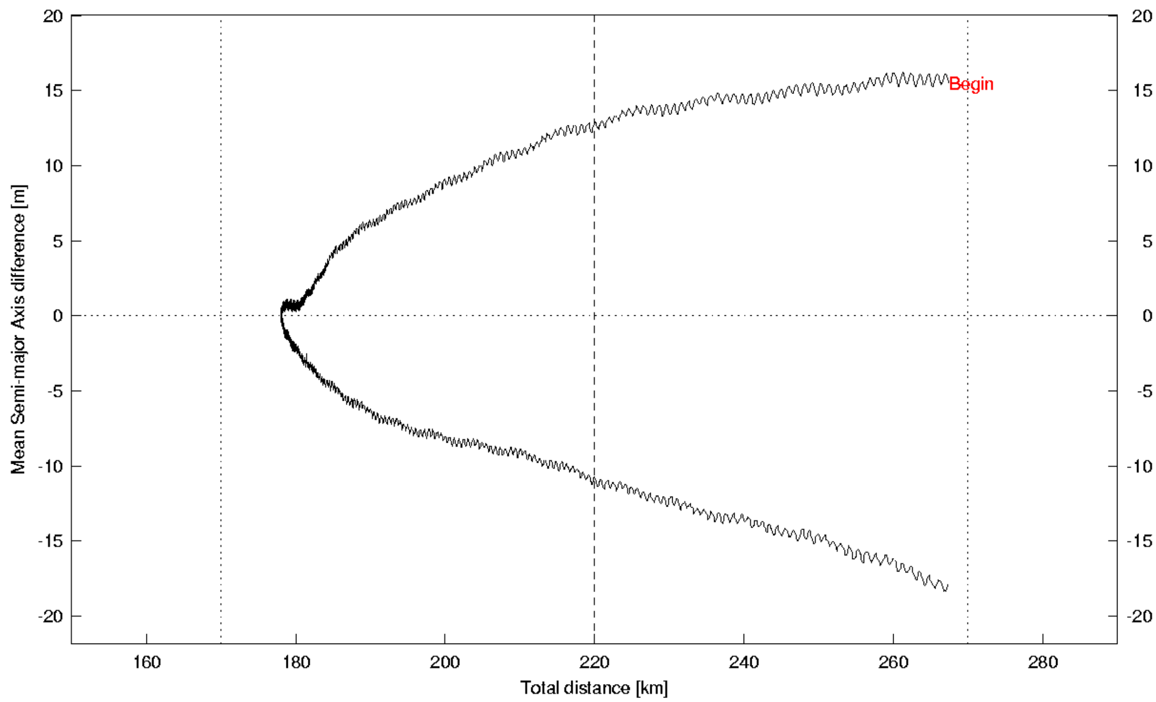

Fig. 4. Numerical simulation of the relative motion of the GRACE spacecraft under the influence of differential drag using realistic solar flux profiles and atmospheric density values as well as ballistic coefficients depending on the varying pitch angle.

RELATIVE POINTING PREDICTION

In order to avoid multipath effects in the K-band radar tracking, the two GRACE s/c must be aligned with an accuracy of 0.5 mrad (corresponding to 110m at a 220km separation). This requires the provision of mutual orbit information to both satellites to allow an on-board computation of the nominal spacecraft orientation in an inertial reference frame as provided by the star sensors. Out of the available options, it has finally been decided to employ ground commanded twoline elements for the prediction of the nominal pointing directions. Compared to ephemeris tables the required upload volume and frequency is tremendously reduced, while the analytical orbit representation requires smaller on-board computer resources and provides longer forecast intervals.

Fig. 5 Difference between the onboard prediction of pitch and yaw angles based on ground commanded twoline elements and the truth values obtained in a numerical trajectory simulation for an altitude of 300 km and for the nominal lower boundary of 170 km over a one week time interval. Due to common error cancellation the onboard prediction stays well within the overall error budget of 0.5 mrads.

Twoline elements for both spacecraft are routinely produced by the GRACE flight dynamics team for ground station support and thus do not imply an additional operations overhead when used for on-board applications. The twoline element sets comprise six orbital elements as well as the ballistic coefficient which allows a modeling of aerodynamic effects on the respective satellite orbit. At the control center, a least-squares fit of pre-computed trajectory data to the SGP4 model is used [7, 8] to derive the seven independent parameters, which are then converted into the traditional twoline representation (comprising two lines of 69 characters) and uploaded to the satellites. On-board each GRACE satellite, the SGP4 model is used to compute inertial positions and velocities of itself and the partner spacecraft from the respective element sets. From these, the line of sight vector in the radial / along-track / cross-track frame and the associated nominal pitch and yaw angles are finally obtained.

Despite the low absolute accuracy of the SGP4 orbit model, on which the twoline elements are based, the relative position perpendicular to the line-of-sight vector can be predicted with extreme accuracy. While individual positions are accurate to between 1 km and 20 km, common error cancellation reduces the errors of the radial and out-of-plane component of the differential position to roughly 0.15 mrad or 30 m (cf. Fig. 5). This leaves a comfortable 2/3 of the available error budget to attitude determination and control errors.

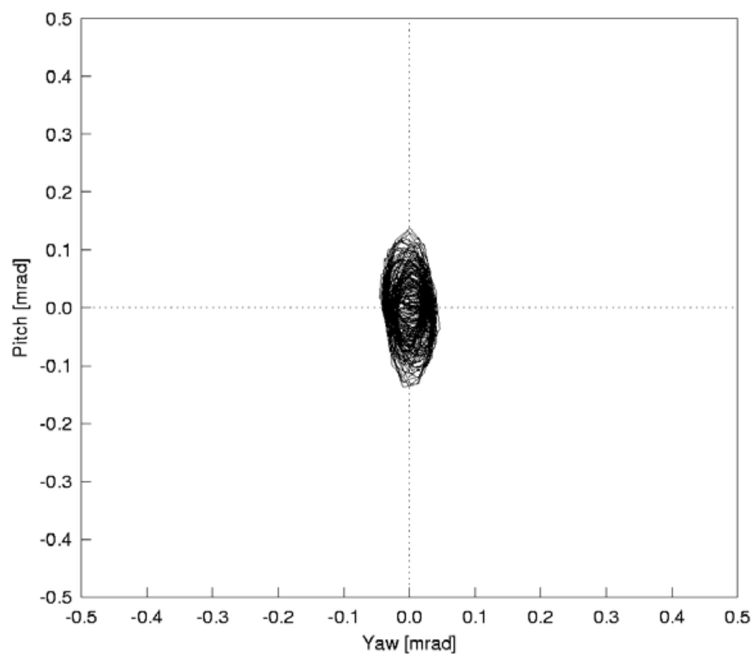

Fig. 5. Difference between the onboard prediction of pitch and yaw angles based on ground commanded twoline elements and the truth values obtained in a numerical trajectory simulation for an altitude of 300 km and for the nominal lower boundary of 170 km over a one week time interval. Due to common error cancellation the onboard prediction stays well within the overall error budget of 0.5 mrads.

SUMMARY AND CONCLUSIONS

The safe and reliable control of the dual satellite GRACE formation poses multiple challenges from a flight dynamics point of view. These include the separation of the two s/c from the upper stage, the initial acquisition and control of the formation in accord with science and mission constraints and the accurate maintenance of the s/c attitude. Suitable strategies for handling these requirements are proposed and assessed.

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